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RESEARCH MEMORANDUM

APPLICATION OF OBLIQUE-SHOCK SENSING SYSTEM TO RAM-JET-ENGINE FLIGHT MACH NUMBER CONTROL

By Fred A. Wilcox and Donald P. Hearth

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

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WASHINGTON
March 3, 1955

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RESEARCH MEMORANDUM

APPLICATION OF OBLIQUE-SHOCK SENSING SYSTEM TO RAM-JET-ENGINE FLIGHT MACH NUMBER CONTROL

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SUMMARY

An oblique-shock sensing system in series with normal-shock positioning controls was investigated on a fixed-geometry, 16-inch ram-jet engine at Mach numbers from 1.5 to 2.16 and angles of attack to 10° in the Lewis 8- by 6-foot supersonic wind tunnel. Position of the oblique shock was sensed by means of a pitot-pressure tube located at the cowl lip. The complete control system was so arranged that the position of the oblique shock was used to provide Mach number control by causing the engine to be operated at high thrust below the desired Mach number and low thrust above the desired Mach number. This system is also applicable to supersonic turbojet-engine installations for control of variable-geometry inlet features.

INTRODUCTION

Ram-jet engines are generally designed to cruise at some predetermined flight speed. To achieve and maintain this Mach number, the control system must sense off-design conditions and provide the necessary corrective action. The conventional pitot-static pressure method is generally employed to determine flight Mach number (ref. 1). Unfortunately, at high altitudes where the static pressure becomes low, this method of Mach number determination may become highly inaccurate.

Among the alternate methods available for sensing the flight Mach number of a ram-jet powered vehicle is a shock sensing system. The position of the oblique shock generated by the spike of an axially symmetric inlet indicates when the design Mach number is reached. Preliminary experiments were undertaken with a 16-inch ram-jet engine utilizing this oblique-shock principle to sense flight Mach number and to regulate the engine thrust accordingly. A brief discussion of this system is included in the preliminary survey of various ram-jet controls (ref. 2). The present report includes detailed results over a range of free-stream Mach numbers of 1.50 to 2.16 and at angles of attack to 10°. Operation of the system investigated is analyzed and various applications are discussed.

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SYMBOLS

The following symbols are used in this report:

C_{F-D}	propulsion thrust coefficient
M	Mach number
P	pitot pressure, lb/sq ft
p	static pressure, lb/sq ft
SFC	specific fuel consumption, (lb/hr)/(lb)
V_1	voltage from pressure switches
V_2	voltage to fuel servo
W_F	fuel flow, lb/hr
$\frac{dW_F}{dt}$	rate of change of fuel flow, $\frac{\text{lb/hr}}{\text{sec}}$
α	angle of attack, deg
ΔP	pitot sensing pressure minus pitot reference pressure, $P_s - P_r$
Δp	static sensing pressure minus static reference pressure, $p_s - p_r$
δ	total pressure divided by NACA standard sea-level static pressure
η	combustion efficiency, percent
θ	total temperature divided by NACA standard sea-level static temperature

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Subscripts:

s	sensing orifice or tube
r	reference orifice or tube
0	free stream
3	combustor inlet

APPARATUS AND PROCEDURE

Engine and Installation

The 16-inch ram-jet engine was installed in the 8- by 6-foot supersonic wind tunnel as shown in figure 1. The supersonic diffuser was designed such that the single oblique shock generated by the 25° half-angle spike would fall on the cowl lip at a free-stream Mach number of 1.9. The plate technique of reference 3 was used to obtain data at test Mach numbers of 2.16. A more detailed discussion of the engine and data reduction techniques is included with engine performance curves in reference 4. Transients in engine nozzle exit area were imposed during control operation by means of a moveable water-cooled plug.

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For convenience in relating the data, the variation of the fuel-flow parameter with combustor-inlet Mach number is given in figure 2 for various values of combustion efficiency. Combustion efficiencies obtained during engine operation ranged from 60 percent to 90 percent and can be obtained from reference 4.

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Location of the pressure tubes and orifices used in the primary and secondary control systems is shown in figure 3 along with the diffuser flow-area distribution. To minimize angle of attack effects, all pressure orifices and tubes were located on the horizontal centerline. The primary, or Mach number sensing system utilized pitot tubes at the cowl lip and at the spike tip. Three pitot-tube positions at the cowl lip were investigated. The secondary control system made use of static-pressure orifices on the engine centerbody and spike as in references 2 and 5.

A can combustor with dual upstream injection was employed (see ref. 4). Fuel flow to the inner fuel manifold was set manually to give a ratio of primary fuel to engine air of about 0.015 for all operating conditions. The fuel to the outer manifold was controlled by the servo-operated fuel valve described in reference 2. This valve was designed to give a linear variation in fuel flow with imposed d-c voltage.

Operation of Control

General arrangement of the control is shown in the block diagram of figure 4(a). Tubes and orifices used to sense shock position were connected to two on-off pressure switches as described in reference 5. Voltage signals from the pressure switches went to a controller which increased or decreased the voltage signal (linearly with time) to the fuel valve. Engine fuel flow thus was varied linearly by the on-off action of the pressure switches. The pressure switches were arranged in series as figure 4(b) indicates. The primary switch which sensed by means of oblique shock position whether flight Mach number was above or below design could either supply a plus voltage signal to the control to increase the fuel flow, or it could set the secondary switch in operation. The secondary switch supplies either a plus or minus voltage signal to the control which would then increase or decrease engine fuel flow to position the normal shock at some prescribed position within the diffuser. For the switch arrangement shown, the secondary switch is held in operation by the primary switch and is giving a signal to the control to increase engine fuel flow. 3567

The primary system was operated on the ΔP signal located between the pitot pressure at the cowl lip and the pitot pressure at the spike tip. This system is illustrated in figure 5. Since the condition for which the oblique shock would intercept the cowl lip is primarily a function of free-stream Mach number, the ΔP signal can be used to indicate whether the engine is operating below or above design flight speed. At flight Mach numbers below design, the cowl pitot tube is behind the oblique shock and pitot sensing pressure P_s is greater than pitot reference pressure P_r. The fuel flow would then be increased to drive the normal shock upstream (fig. 5(a)). As the slip line (vortex sheet) crosses the cowl-pitot tube, the pitot sensing pressure is approximately equal to the pitot reference pressure and the secondary control system would be activated to reduce the fuel flow from the condition shown. Thus, at Mach numbers below the design value of 1.9, the combined systems position the normal shock so that the slip line, generated at the intersection of the oblique and the normal shocks, intersects the cowl lip (fig. 5(b)). At flight Mach numbers above the design value, the secondary shock positioning control system would be continually activated since the sensing and reference pitot pressures would be approximately equal (fig. 5(c)).

The secondary control system when activated by the primary or Mach number sensor system sets the normal shock at some predetermined position in the diffuser. Two positions for the normal shock were investigated and are illustrated in figures 6 and 7. For the sensing pressures shown in figure 6 the normal shock was positioned approximately 15 inches downstream of the cowl lip. This downstream or supercritical location resulted in low diffusor pressure recovery (ratio of diffusor-exit total pressure to free-stream total pressure) which in turn caused low engine thrust. Thus, when used in series with the primary system, high thrust would be set at flight Mach numbers below design and reduced thrust for Mach numbers above design. Normal-shock position close to the cowl lip (critical) was obtained by use of the sensing pressures shown in figure 7. This arrangement of the secondary system was the same utilized in reference 5, and when combined with the primary system results in essentially peak thrust operation over the Mach number range investigated.

RESULTS AND DISCUSSION

To determine the optimum position for the pitot tube located at the cowl and utilized in sensing the position of the oblique shock generated by the spike, three alternate locations were investigated. The engine was operated manually and the observed variation in pressure parameter for each tube position is indicated in figure 8. Although shown only at a Mach number of 1.50, the results were consistent at other Mach numbers below 1.9. Ideally, the pitot-pressure parameter should be maximum and constant as the inlet normal shock is driven forward until the vortex sheet passes across the pitot tube at the cowl lip. Once the vortex sheet passes across the sensing tube and enters the inlet, the pressure parameter should ideally decrease to zero. The band of data indicated in figure 8 for a given tube location is probably due in part to the finite thickness of the vortex sheet. Because some gain in propulsive thrust may be achieved by subcritical operation (ref. 4), a tube located 0.110 inch outside the cowl lip was selected for use with the primary control system. The dashed lines on the figure indicate the setting selected for the pressure switch utilized as part of the primary control system.

The effect of free-stream Mach number on the pitot pressure P_s for the tube position 2 is indicated in figure 9. These data, which were also obtained during manual engine operation, indicate that as the free-stream Mach number is increased from 1.50 to 1.79, the engine operation remains subcritical. The value of the pressure parameter for supercritical operation increased with flight Mach number because of increased strength of the oblique shock.

For Mach number above design ($M_0 > 1.9$), a condition for which the oblique shock would fall inside the cowl lip, the sensing pressures were identical to that presented for Mach number 2.16, that is, constant and approximately zero. Generally good agreement was noted between the supercritical pitot-pressure parameters observed experimentally and the values predicted analytically from shock structures.

The static-pressures used to actuate the supercritical secondary system are illustrated in figure 10. Data for free-stream Mach numbers of only 1.98 and 2.16 are shown since this system was to be placed in operation only above design ($M_0 = 1.9$). The static-pressure parameters for Mach numbers below design are similar to those presented for Mach numbers above design. The horizontal dashed lines represent the setting selected for the pressure switch. The data indicate, that as desired, this control system would set a supercritical operating condition over the range of Mach numbers investigated and at angles of attack to at least 10°. Although not indicated on figure 10, good agreement was obtained at Mach number 1.98 between the experimental static-pressure parameter and values computed from one-dimensional flow relations. Similar agreement was also observed at, and below, the design Mach number. Poor agreement, however, was noted at 2.16, where the oblique shock fell inside the cowl.

Static pressures for the critical secondary system are presented in reference 5. These pressure data as well as test results on the actual control system, also reported in reference 5, indicated that the system should and did maintain the engine slightly subcritical. It was also demonstrated that this control system operated satisfactorily at angles of attack to 10°, the maximum investigated. 3567

Engine performance set by the control is shown in figure 11, superimposed on the steady-state engine performance curves originally presented in reference 4. Data are shown for the primary system alone and in series with the supercritical and critical secondary systems. In addition, data are presented for the supercritical secondary system which can also operate independently. The data presented are for low rates of change of fuel flow which gave fuel-flow oscillations below 130 pounds per hour, or about $3\frac{1}{2}$ percent of the total fuel flow. Consequently, errors due to time averaging of pressures used in computing the data are minimized. A discussion of dynamic performance of the combined control systems is presented in the appendix.

At Mach numbers of 1.5 and 1.79, the oblique shock fell ahead of the cowl lip, and the primary system by itself and in series with either of the secondary systems, set a subcritical inlet condition close to what would be expected from analysis of the sensing pressures (see fig. 9). This subcritical operating condition resulted in close to peak propulsive thrust. At all Mach numbers the supercritical secondary system set a drastically reduced engine thrust. This low thrust would be obtained for Mach numbers above 1.9 when this secondary system was used in series with the primary system. The critical secondary system, when used in series with the primary system, set a slightly subcritical operating point at free-stream Mach number 1.98 and a near critical point at free-stream Mach number 2.16, both resulting in nearly peak propulsive thrust.

The oblique-shock sensing system reported herein may also be applied to ram-jet engines, which should not be operated subcritically because of possible combustor blow-out due to diffuser instability. Instead of requiring subcritical inlet operation as in the present test, an additional normal-shock sensing switch could be activated by the oblique-shock sensor such that a slightly supercritical operating point would be obtained for that part of the flight plan when a relatively high thrust condition would be required.

Control of the flight Mach number of a missile by use of the primary and low-thrust secondary systems in series is summarized in figure 12. Engine propulsive thrust coefficient and engine combustion-chamber inlet Mach number are plotted as a function of free-stream Mach number. The free-stream Mach number at which the controlled thrust dropped to the lower level was determined as 1.91 $\pm$0.01 by slowly changing free-stream Mach number by adjusting the tunnel throat area. This spread in the controlled-flight Mach number is approximately the Mach number spread required for the oblique shock to pass from one side of the inside diameter of the pitot sensing tube to the other. The tube used had an inside diameter of 0.078 inch. The spread could be reduced by flattening the tube or by going to larger scale engines.

This combined control system would provide speed control over the Mach number range tested if the drag coefficient of the missile falls within the propulsive thrust range available from the engine. For the hypothetical missile drag shown in figure 12, the primary control system would call for excess thrust if the flight Mach number fell below 1.90 and the missile would accelerate. When the flight Mach number reached 1.92, the secondary system would be activated and the missile would decelerate. The missile speed would thus oscillate above and below the design flight speed.

Although this system is effective in modulating thrust and maintaining the design free-stream Mach number it is not necessarily efficient when applied during the boost phase of flight. More efficient boost operation may be obtained by utilizing variable-geometry features such as a translating-inlet spike or a variable-throat area exit nozzle (ref. 6). Desired control over such variable-geometry features can be achieved with the primary or oblique-shock sensing system discussed herein. An example of such an application is presented in reference 7, which discusses the control of spike translation of a supersonic inlet-turbojet engine combination.

It is interesting to compare the potential accuracy of an oblique-shock sensing system in determining when a desired flight Mach number is reached, and the method of static-to-pitot pressure ratio generally employed. Considering only the signal strength available to each method at 80,000 feet of altitude ($p_0 = 58$ lb/sq ft abs), the trends shown in figure 13 are obtained. The design Mach number selected was 2.8. Utilizing a 28° half-angle cone, a pressure differential of over 400 pounds per square foot would be available to the oblique-shock sensing system (fig. 13(a)). More efficient inlet types having a higher pressure recovery would provide an even greater pressure differential.

The technique of determining flight Mach number from the measured static-to-pitot pressure ratio is illustrated in figure 13(b). Because of the low static-pressure values at high altitudes, a small error in measurement can result in large Mach number errors. For example, a ±5 pounds per square foot error in the static-pressure determination results in an error of ±0.12 in the computed flight Mach number, as indicated by the cross-hatched area in figure 13(b).

SUMMARY OF RESULTS

From an investigation in the Lewis 8- by 6-foot supersonic wind tunnel of an oblique-shock sensing control system on a 16-inch ram-jet engine at Mach numbers from 1.5 to 2.16 and angles of attack to 10° the following results were obtained. 3567

1. Accurate determination of the relation between oblique-shock position and flight Mach number was obtained by measuring the pitot pressure at the cowl lip. A large pressure differential is available to sense the oblique-shock position even at high altitudes.

2. When used in conjunction with a normal-shock positioning control system the combination will provide speed control for a fixed-geometry ram-jet powered vehicle.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, December 20, 1954

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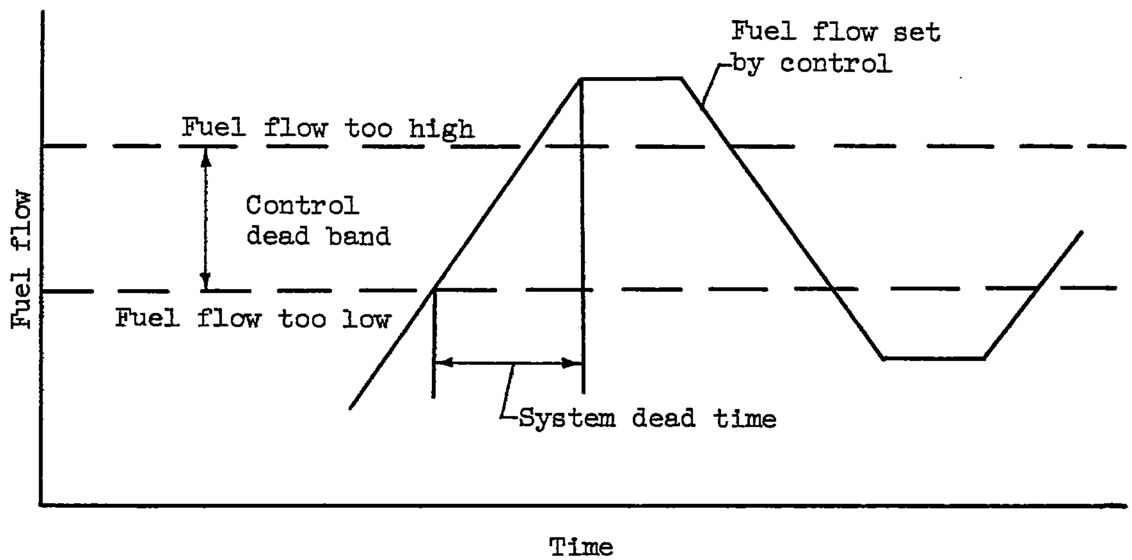
APPENDIX - DYNAMIC PERFORMANCE OF CONTROL

Dynamic performance of the critical secondary system is presented in reference 5. Because the dead bands of the supercritical and critical secondary systems were both nearly zero, similar dynamic performance was obtained for both systems. Because of the wide dead band for the primary system, as shown in figure 9, the dynamic performance differed from that of the two secondary systems. Oscillation of this system would be expected to follow the flat top wave shape described in reference 5, and shown in the following sketch:



This type of control oscillation is illustrated in the oscillograph trace of figure 14(a). This trace is for the primary system at free-stream Mach number 1.79. The rate of change of fuel flow used was 10,700 pounds per hour per second. The system dead band for this condition was from combustor-inlet Mach number 0.176 to 0.188 (see fig. 9(b)) which in terms of fuel flow was about 300 pounds per hour. The measured frequency of oscillation for this trace is 4.2 cycles per second. From equation (1) of reference 5, the system dead time is calculated to be 0.0595 seconds. This value is in approximate agreement with the value of 0.055 seconds obtained by summing the component dead times (ref. 5). Controlled fuel flow oscillated between 1740 pounds per hour and 2700 pounds per hour, giving a total amplitude of oscillation of 960 pounds per hour. Average total engine fuel flow for this case was 3647 pounds per hour, 1427 pounds per hour being manually set by the primary fuel system. Predicted amplitude of oscillation from equation (2), reference 5, was 972 pounds per hour.

It was indicated in reference 5, that no oscillation of this type of on-off control would be obtained at rates of change of fuel flow below a critical value that depends on the dead band width. The trace shown in figure 14(b) illustrates a case where an initial oscillation, resulting when the control was turned on, was damped because a sufficiently low rate of change was used. The trace is for the primary system at free-stream Mach number 1.6 and rate of change of 7150 pounds per hour per second. Unlabeled lines on this trace are not pertinant to the discussion. The approximate, experimentally determined value of the system dead band for this this condition was 400 pounds per hour. 3567 Using this value of dead band and a system dead time of 0.0595 seconds, the critical value of the rate of change is calculated to be $\leq$ 6720 pounds per hour per second from equation (4) of reference 5. The difference between this value and the rate of change used for the trace is within the accuracy of determination of dead-band width.

In figure 15(a), the fuel flow was manually displaced at free-stream Mach number 1.90 to a total value of 4577 pounds per hour, a farther subcritical condition than that set by the control. Total fuel flow for critical operation at this Mach number was about 3000 pounds per hour. This trace is for the primary system in series with the critical secondary system and the rate of change of fuel flow used was 7150 pounds per hour per second. Since the free stream Mach number was slightly below the control value of 1.91, the control set a slightly subcritical engine operating condition at which the fuel flow oscillated between 3320 and 3790 pounds per hour with a flat-top wave shape.

Figure 15(b) shows the operation of the primary system at an free-stream Mach number of 1.79 during a change in exit area from 0.96 square feet to 0.76 square feet in about 0.07 second by means of the water cooled exit plug. Control action reduced the initial total fuel flow of 3405 pounds per hour to an average total fuel flow of 2297 pounds per hour. The dead band for this case was sufficiently wide that no regular oscillation of the fuel flow was obtained, but instead the total fuel flow drifted irregularly between 2060 and 2535 pounds per hour.

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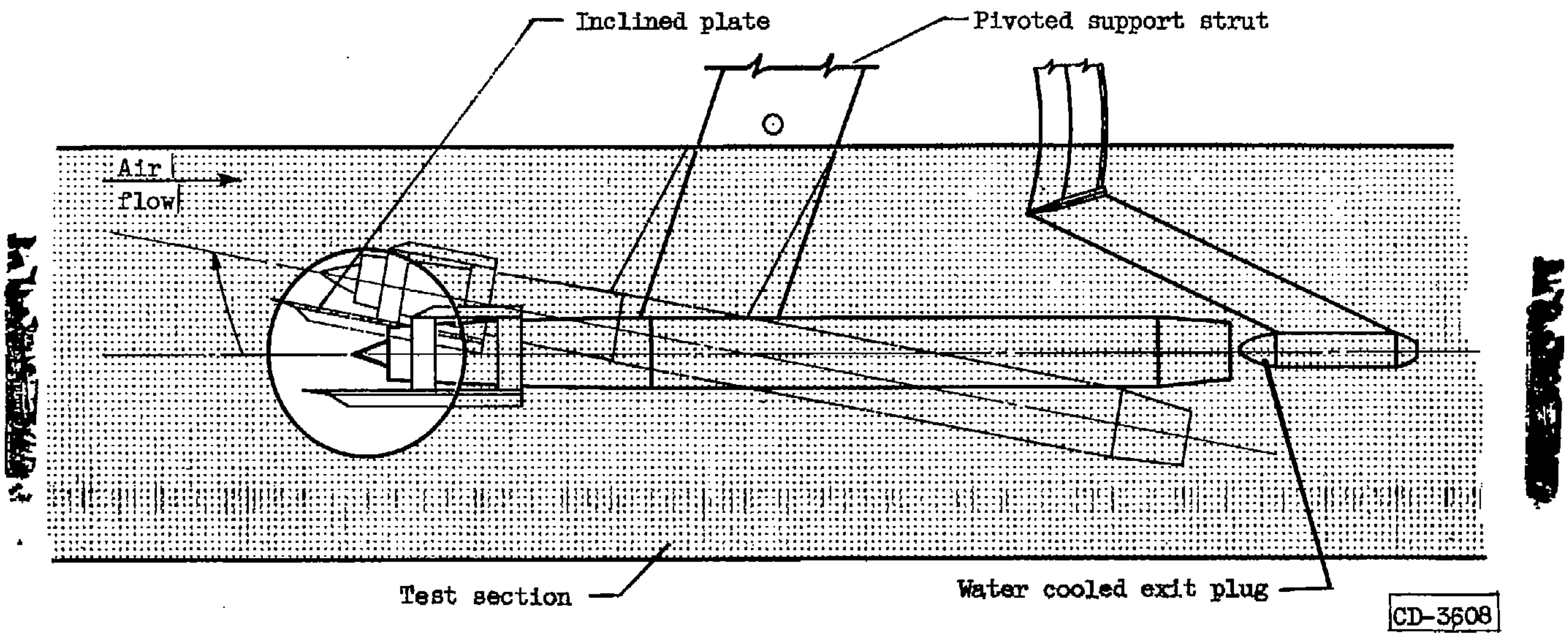


Figure 1. - Installation of ram-jet engine in 8- by 6-foot supersonic wind tunnel.

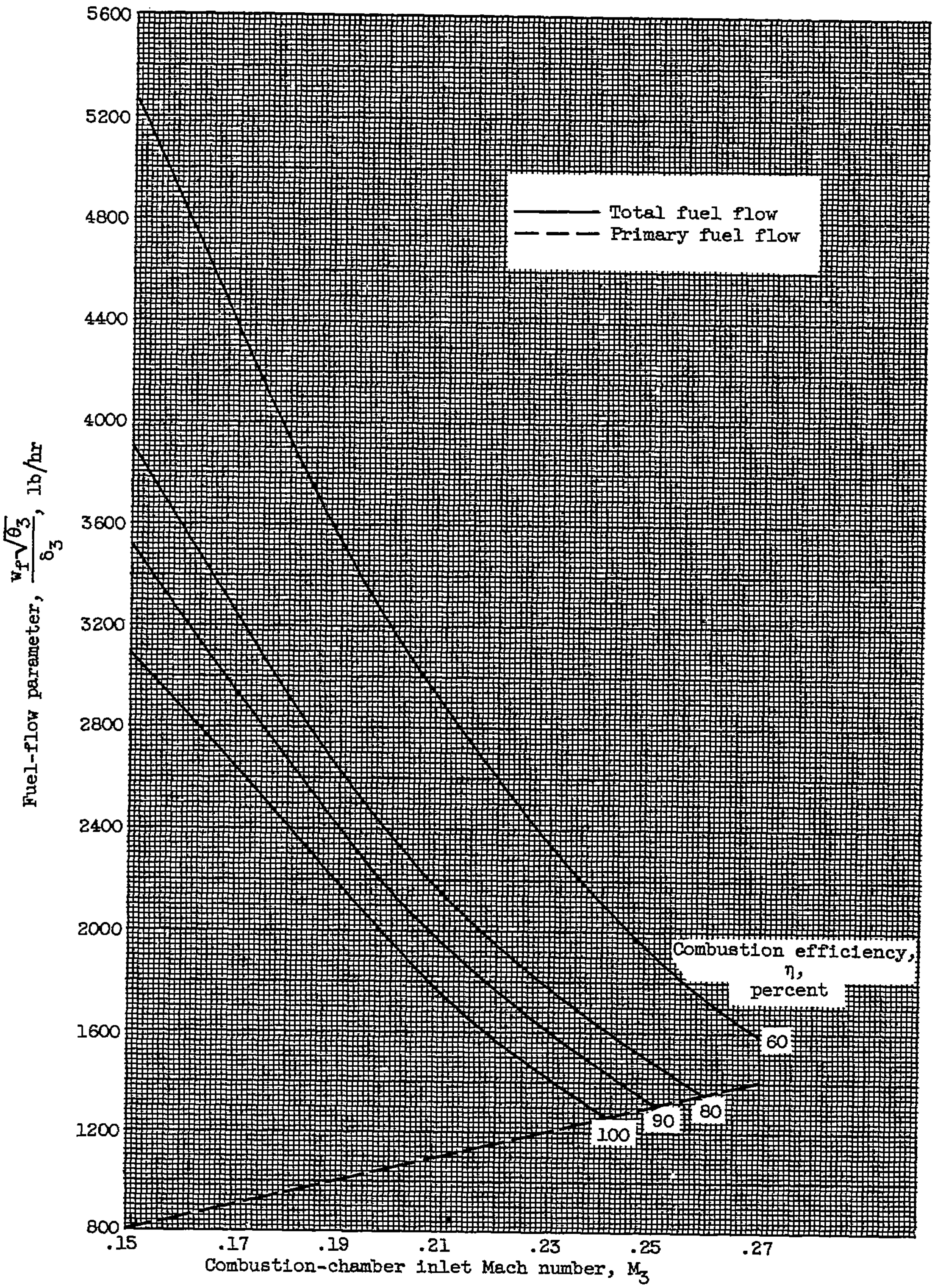


Figure 2. - Engine fuel-flow characteristics.

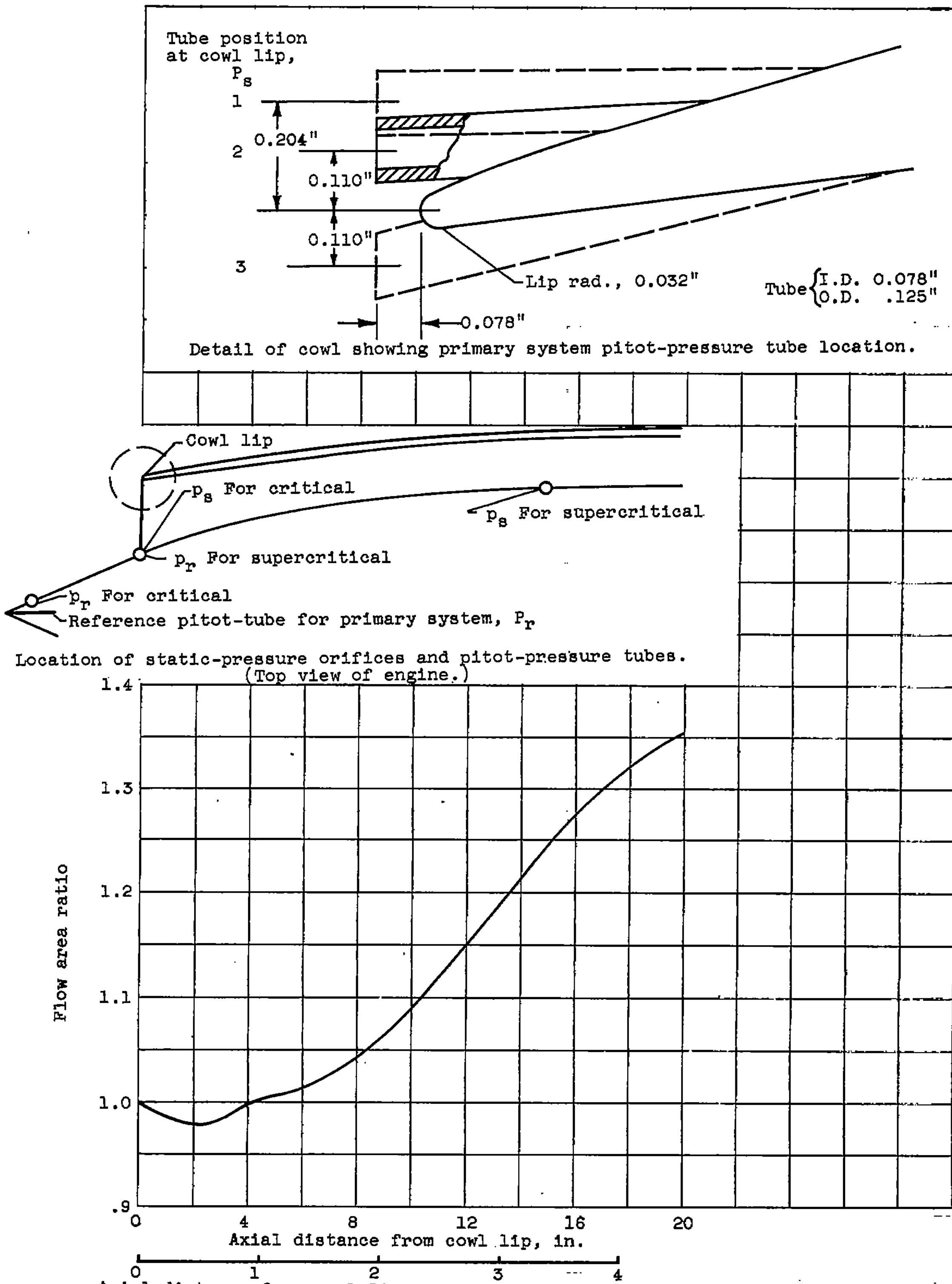
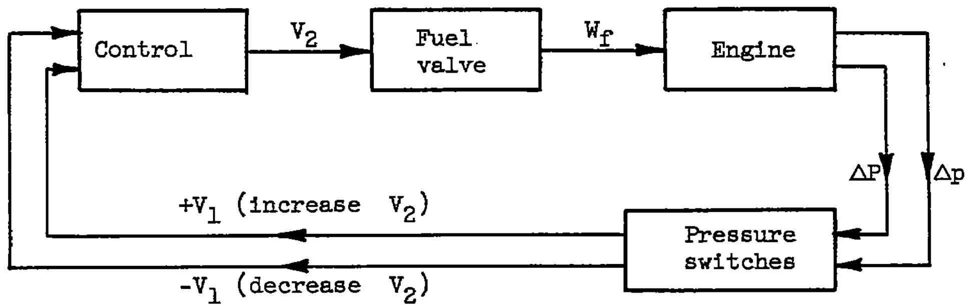
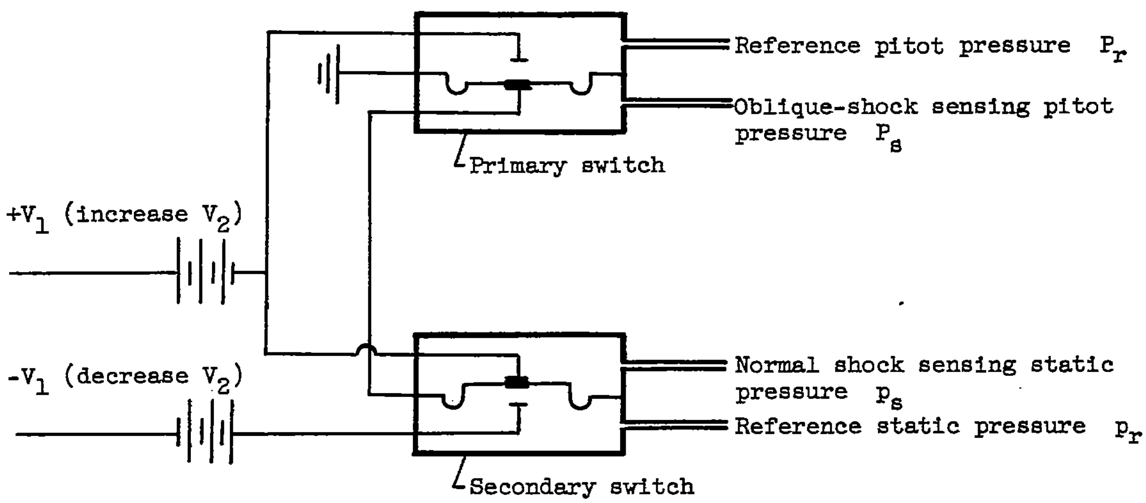


Figure 3. - Diffuser area variation and tube locations.



(a) Block diagram.



(b) Pressure switch arrangement.

Figure 4. - Block diagram and pressure switch arrangement of control system.

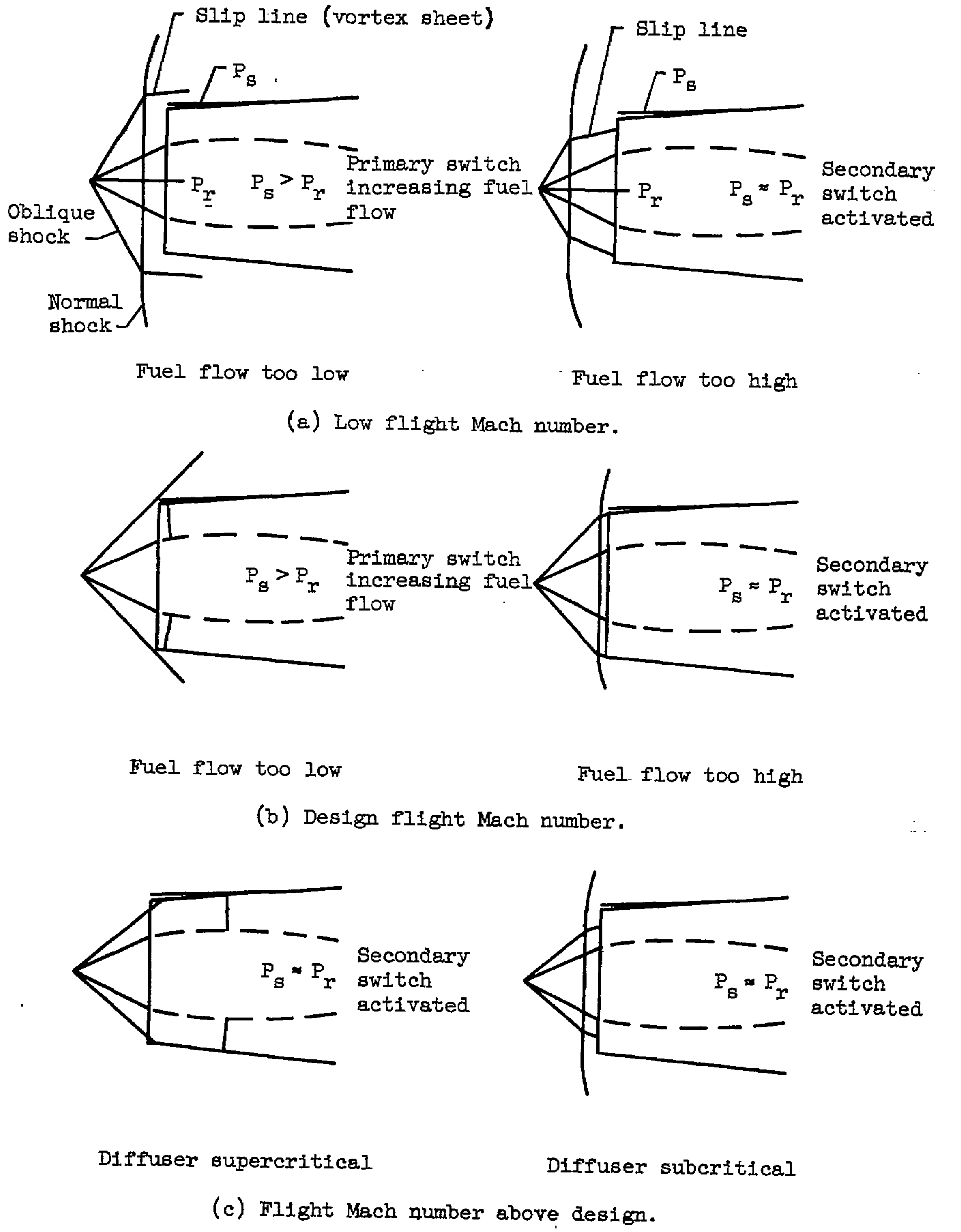
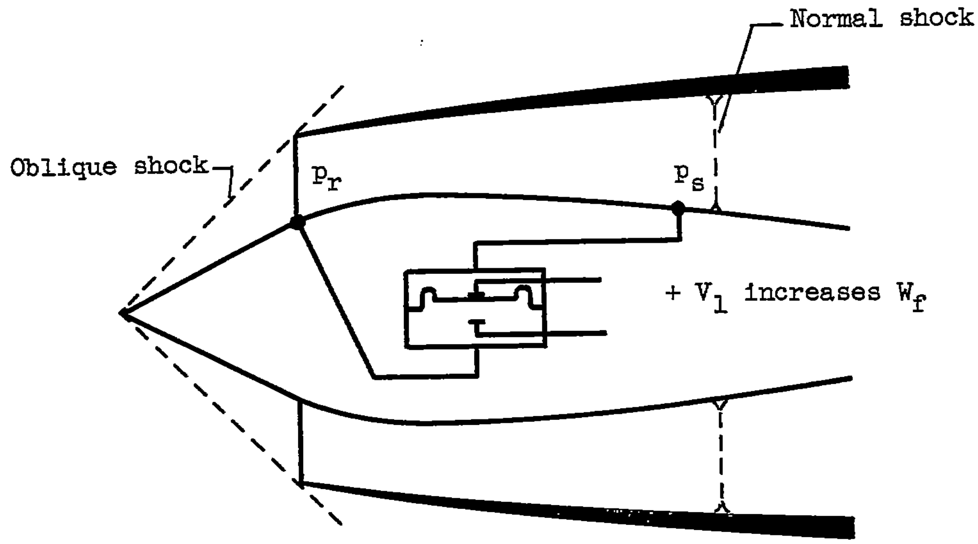
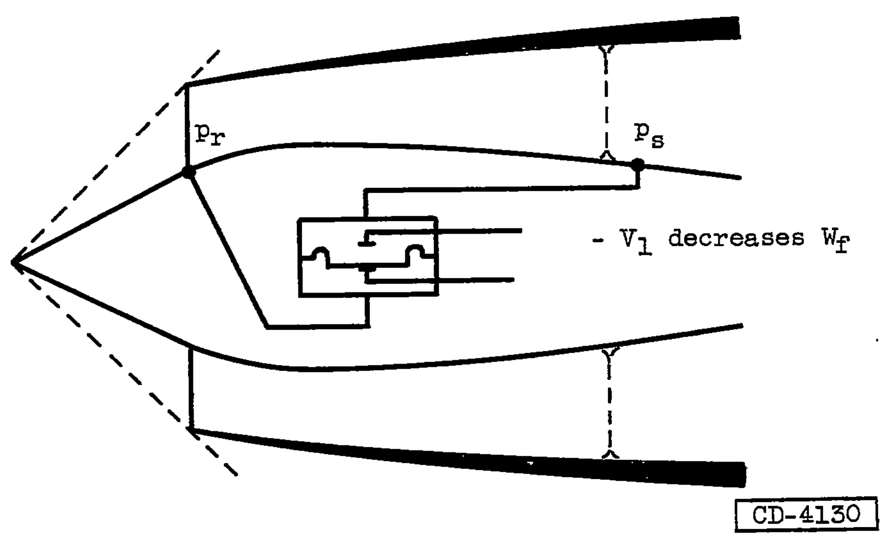


Figure 5. - Schematic diagram of operation of oblique-shock sensing system.



$p_s < p_r$

(a) Fuel flow too low.



$p_s > p_r$

(b) Fuel flow too high.

Figure 6. - Schematic diagram of operation of supercritical secondary system.

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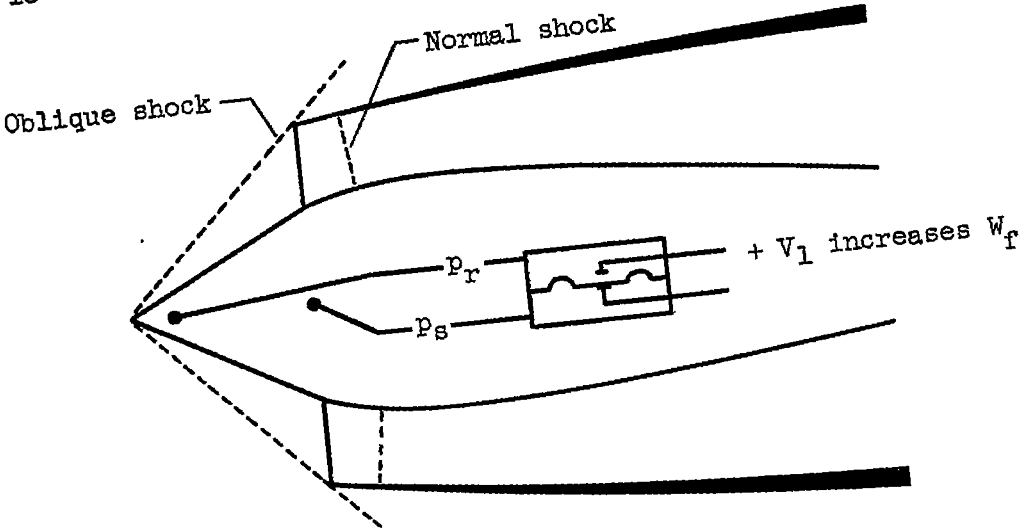
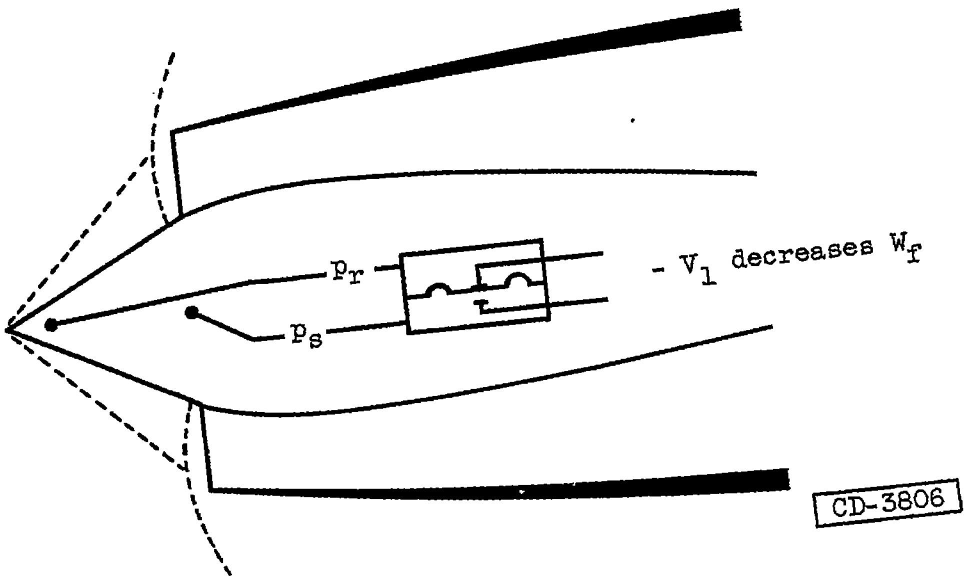


$p_x \approx p_r$

(a) Fuel flow too low.



$p_s > p_r$

(b) Fuel flow too high.

Figure 7. - Schematic diagram of operation of critical secondary system.

[redacted]

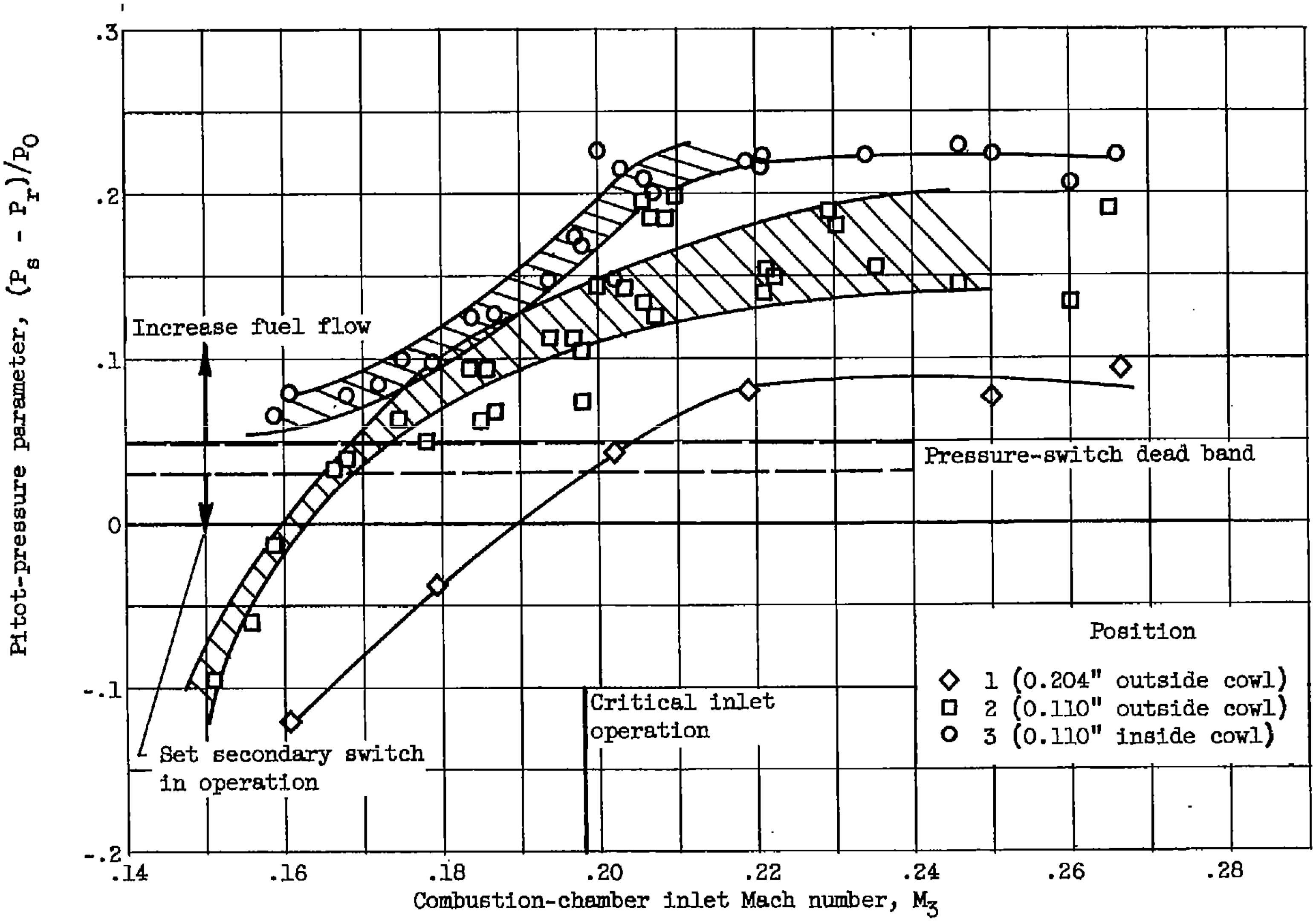
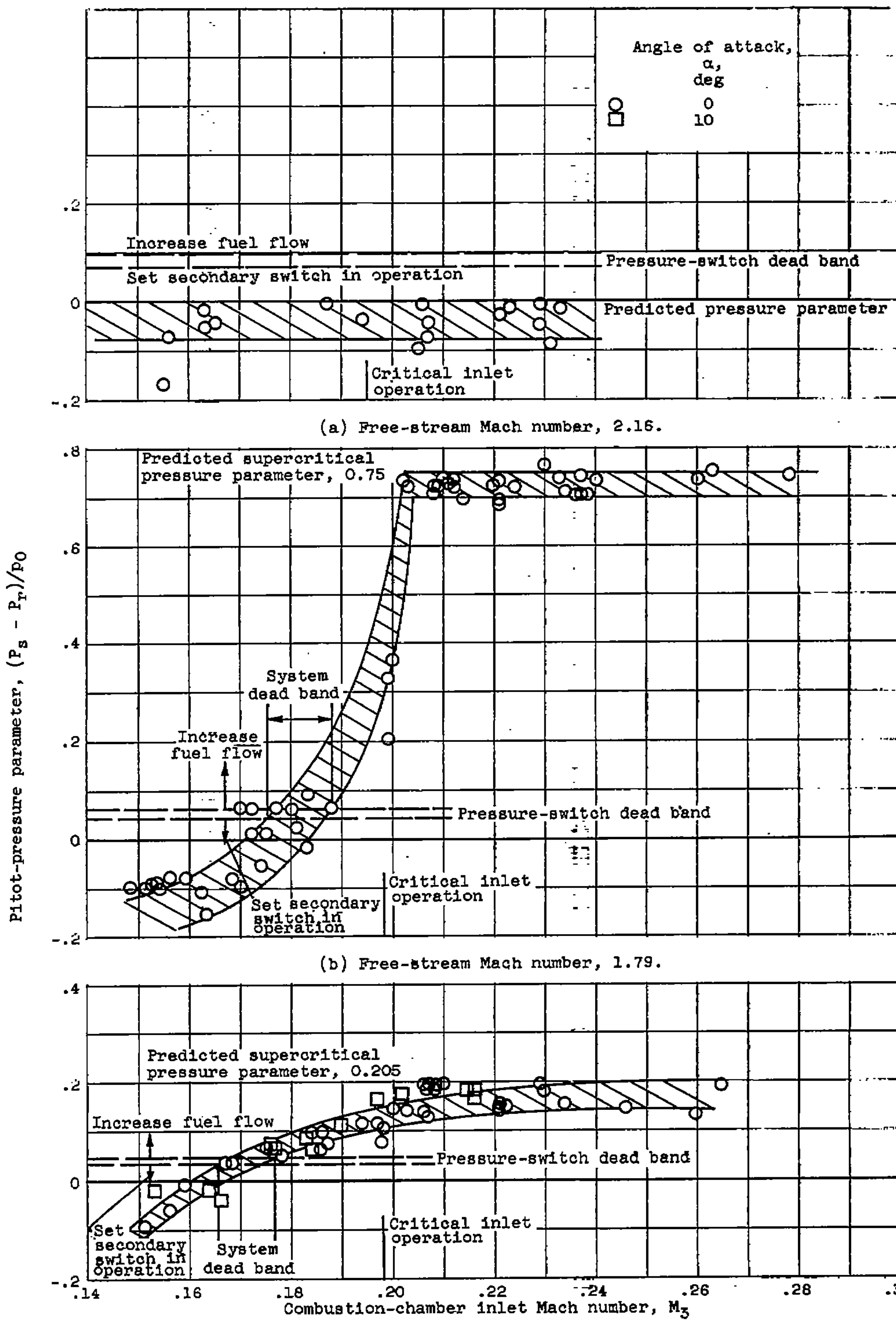
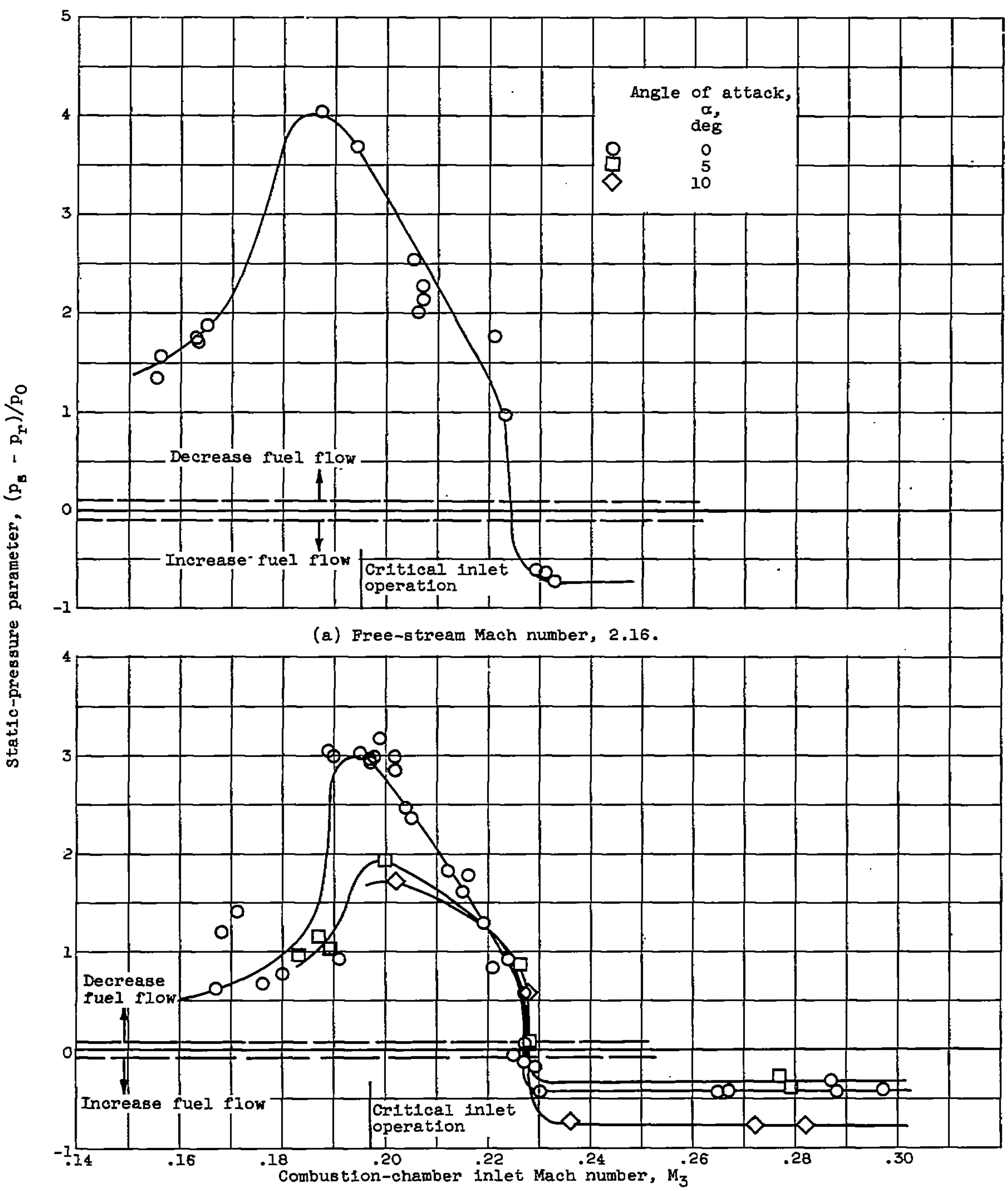


Figure 8. - Effect of tube location on sensing pressure for primary system. Free-stream Mach number, 1.5; angle of attack, 0°.



(c) Free-stream Mach number, 1.5.

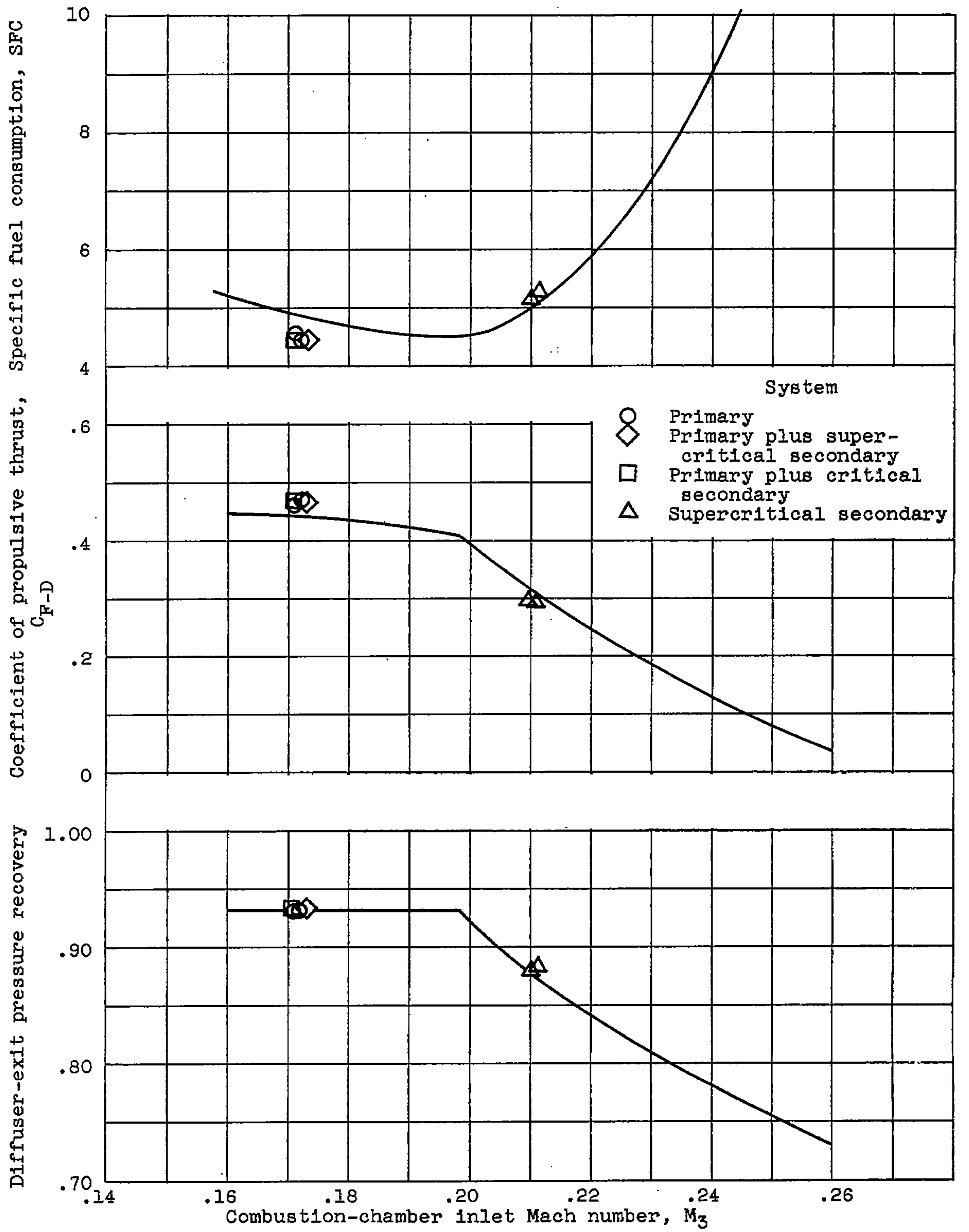
Figure 9. - Effect of Mach number on sensing pressures for primary system tube position 2.



(a) Free-stream Mach number, 2.16.

(b) Free-stream Mach number, 1.98.

Figure 10. - Static-pressure parameters for supercritical secondary system.



(a) Free-stream Mach number, 1.50; angle of attack, 0°.

Figure 11. - Steady-state performance set by control. Data superimposed on steady-state performance curves of reference 4.

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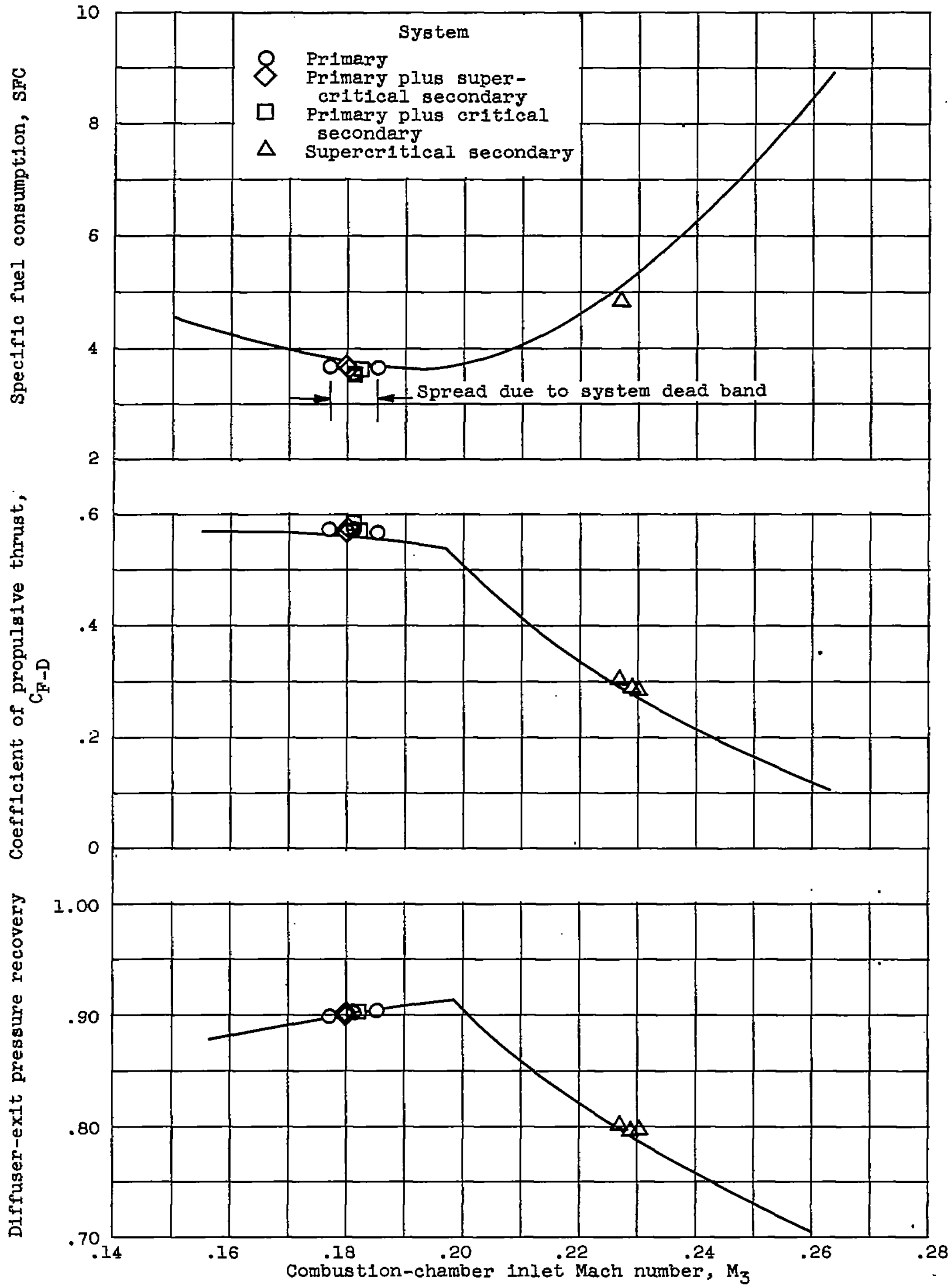
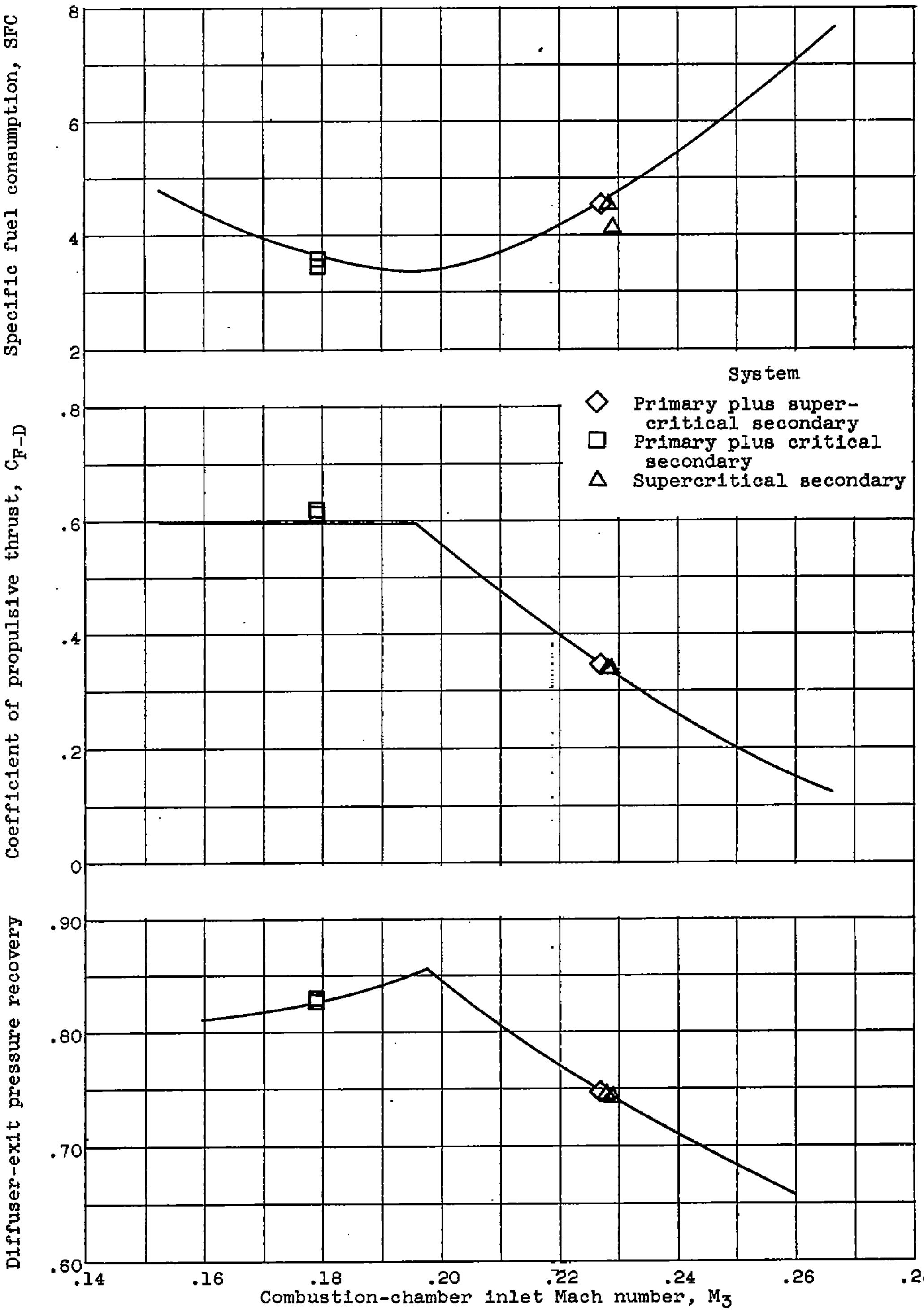


(b) Free-stream Mach number, 1.79; angle of attack, 0°.

Figure 11. - Continued. Steady-state performance set by control. Data superimposed on steady-state performance curves of reference 4.



(c) Free-stream Mach number, 1.98; angle of attack, 0°.

Figure 11. - Continued. Steady-state performance set by control. Data superimposed on steady-state performance curves of reference 4.

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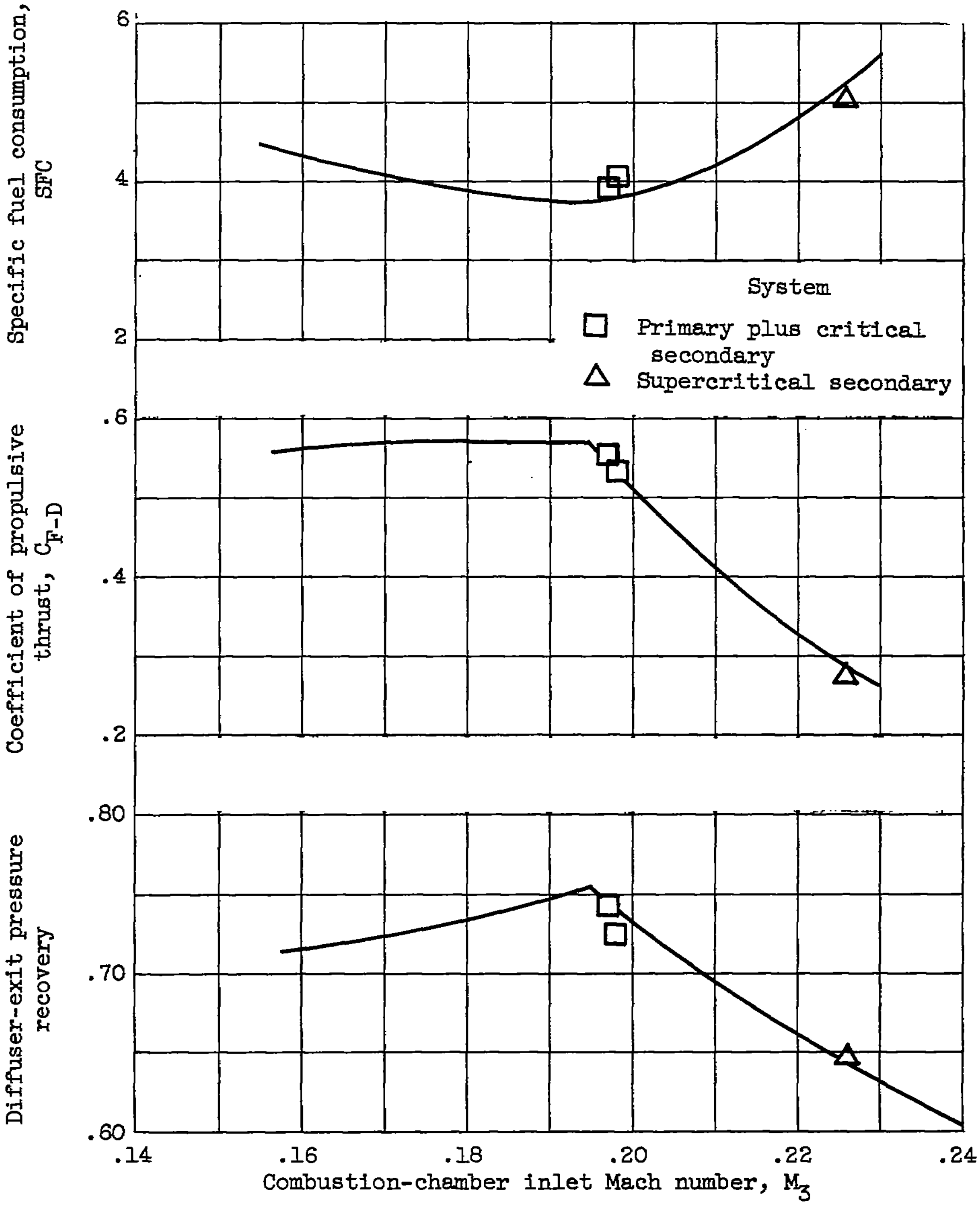


(d) Free-stream Mach number, 2.16; angle of attack, 0°.

Figure 11. - Concluded. Steady-state performance set by control. Data superimposed on steady-state performance curves of reference 4.

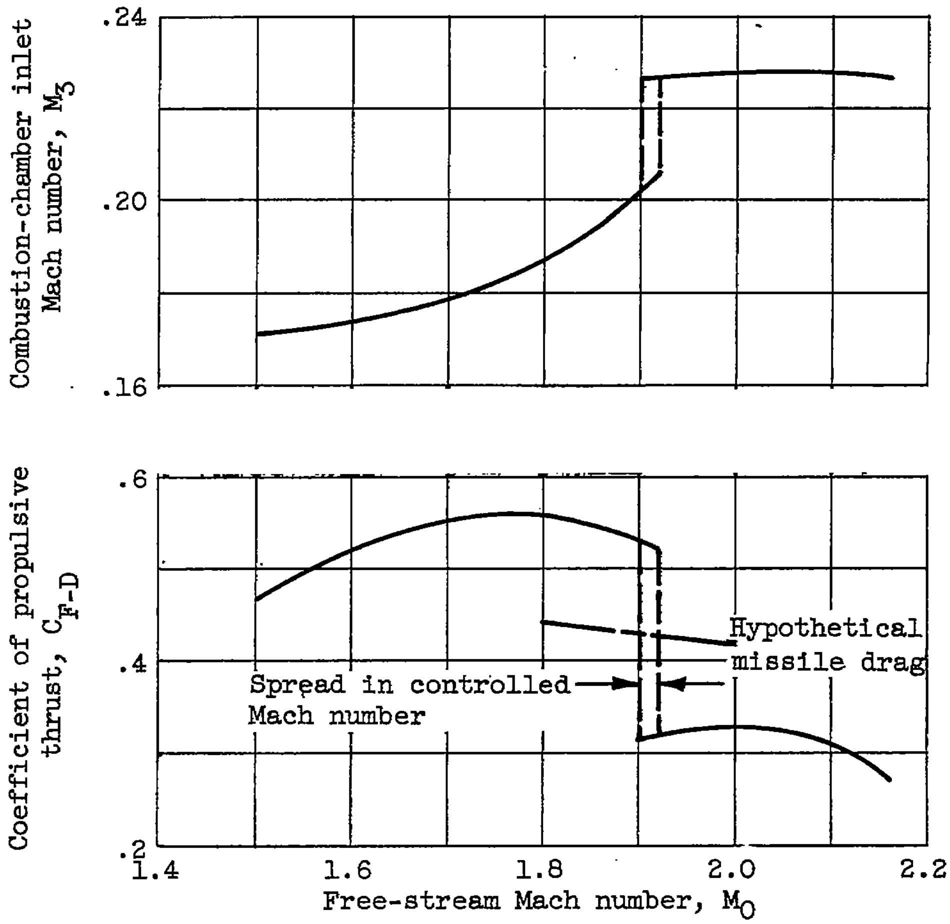
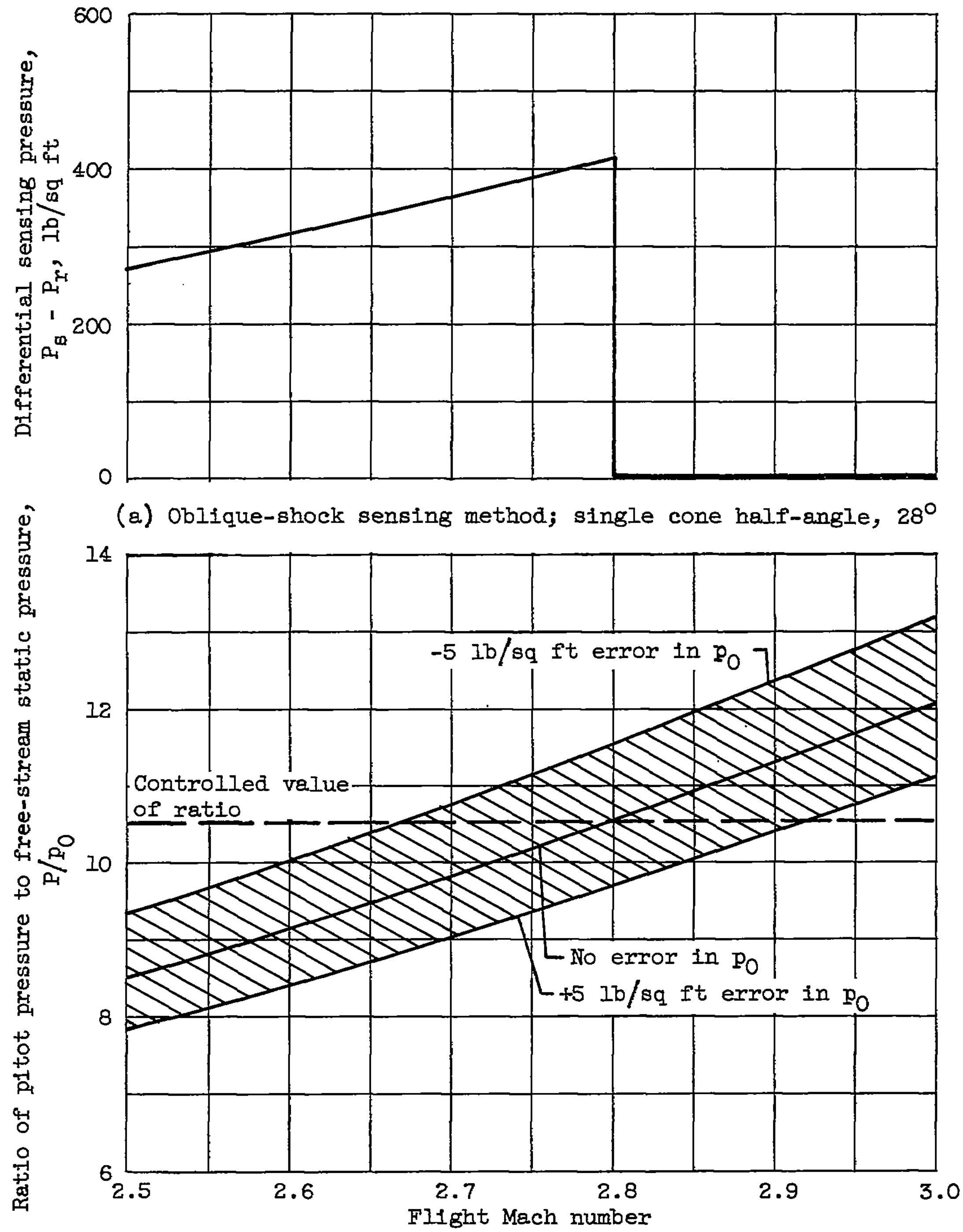
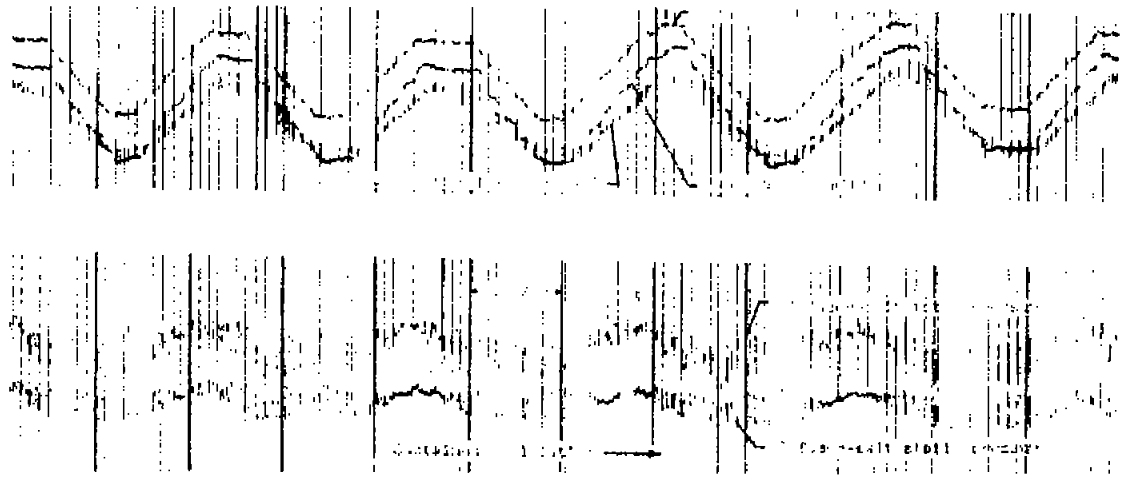


Figure 12. - Effect of free-stream Mach number on controlled engine performance for primary plus supercritical secondary systems.

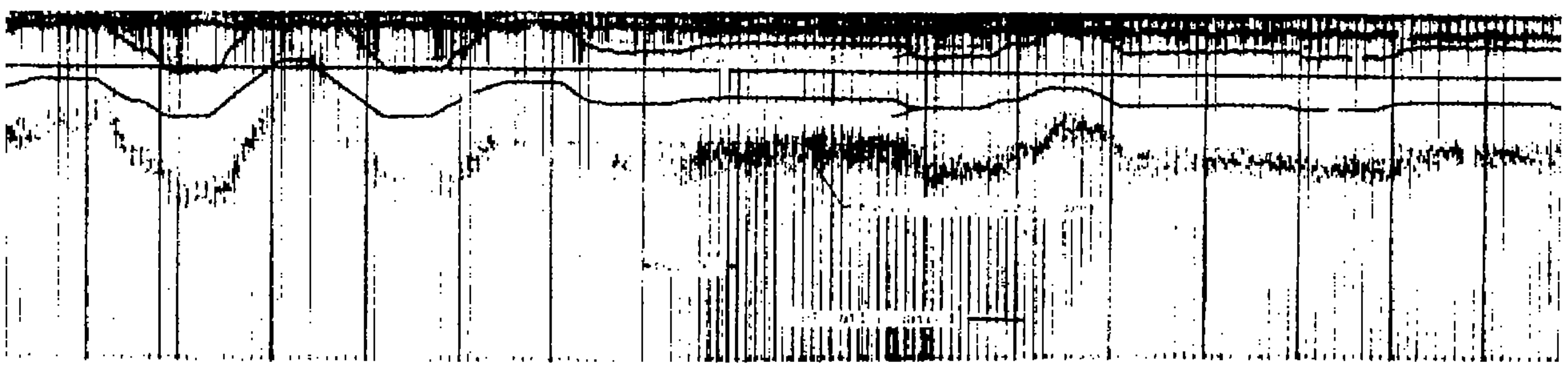


(b) Pressure ratio method.

Figure 13. - Estimated pressures available for controlling flight Mach number at 2.8 at 80,000 foot altitude.

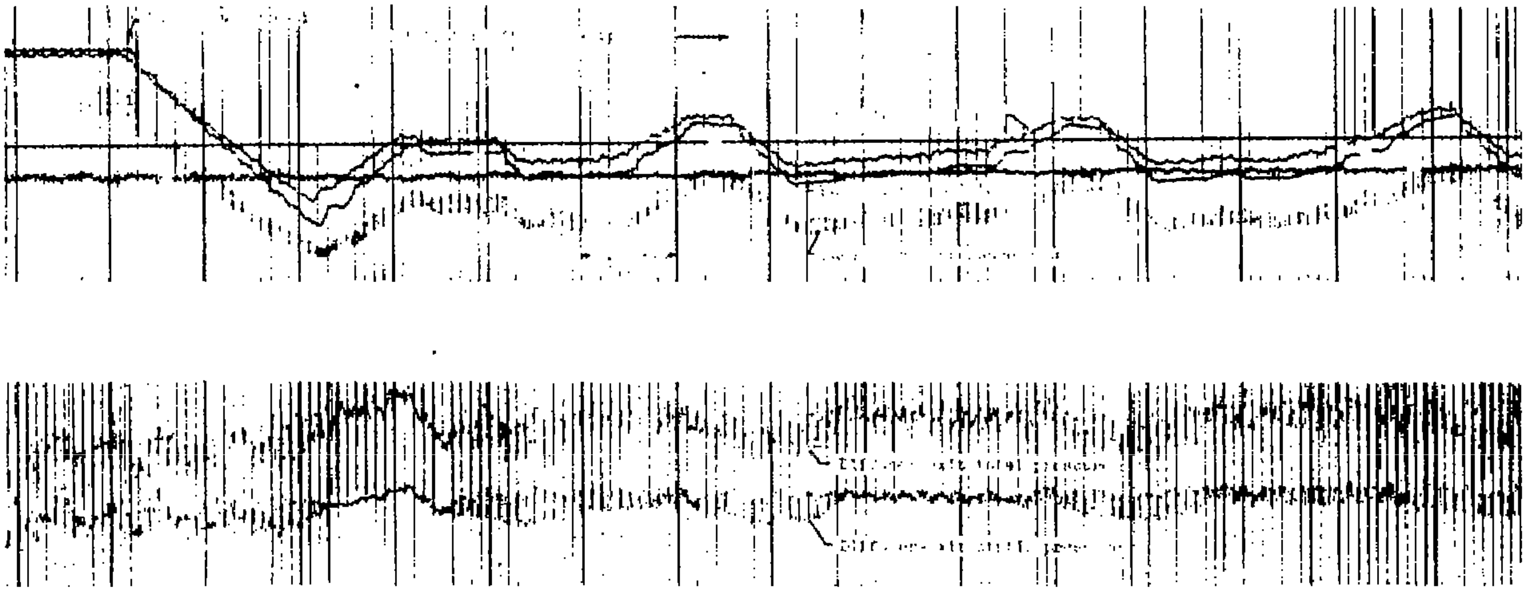


(a) Free-stream Mach number, 1.79; rate of change of fuel flow, $\frac{dW_f}{dt}$, 10,700 pounds per hour per second; frequency of oscillation, 4.2 cycles per second.

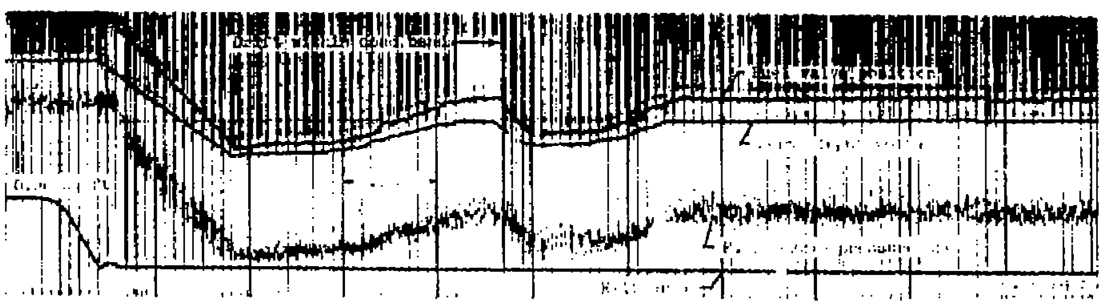


(b) Free-stream Mach number, 1.6; rate of change of fuel flow, $\frac{dW_f}{dt}$, 7150 pounds per hour per second; initial frequency of oscillation, 4.1 cycles per second.

Figure 14. - Typical traces taken during primary control oscillation.



(a) Fuel flow displaced manually to subcritical operating condition. Free-stream Mach number, 1.9; rate of change of fuel flow, $\frac{dW_f}{dt}$, 7150 pounds per hour per second; oscillation frequency, 2.9 cycles per second.



(b) Exit area reduced. Free-stream Mach number, 1.79; rate of change of fuel flow, $\frac{dW_f}{dt}$, 7150 pounds per hour per second.

Figure 15. - Operation of primary control during imposed transients.

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